Project 101072599 — HORIZON-MSCA-2021-DN-01

Views and opinions expressed are however those of the author only and do not necessarily reflect those of the European Union or EUROPEAN RESEARCH EXECUTIVE AGENCY. Neither the European Union nor the granting authority can be held responsible for them.

Funded by the European Union

Damage and failure in composite material and structures

Gustavo Baños Lapuente
Airbus Defence and Space

April10th 2024

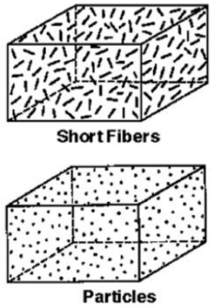
Contents

USES 2 Funded by the European Union

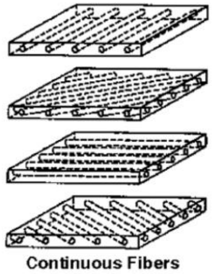
Introduction to mechanics of composite materials (i)

Fibrous composites and the orthotropic ply

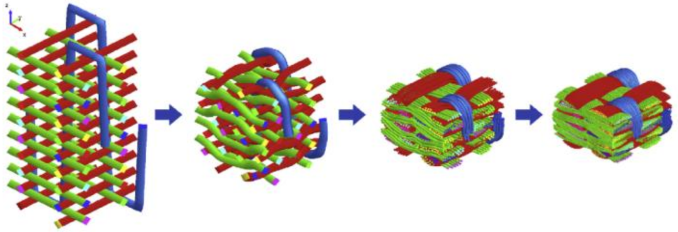
Reinforcement: short fibres

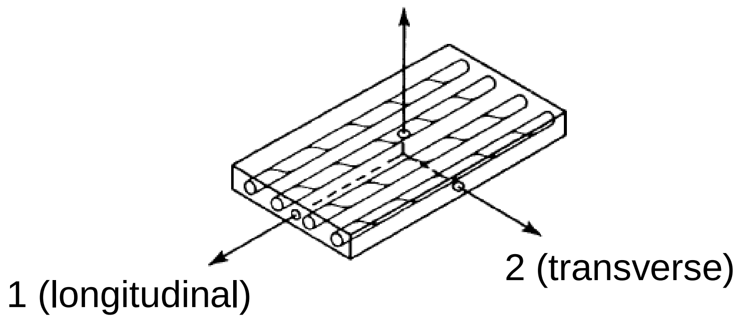


Reinforcement: Continuous fibres

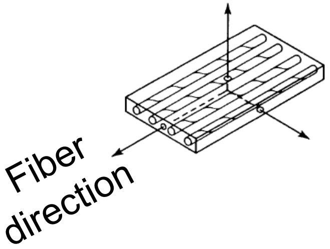


3D woven composite

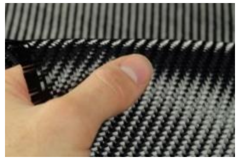


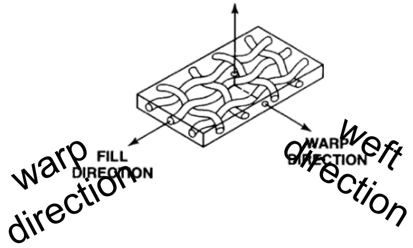


UD-Tape

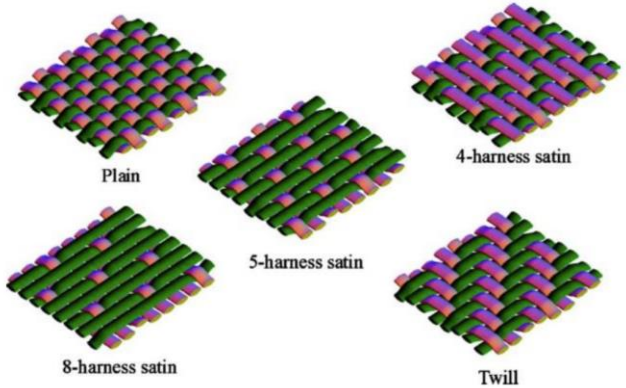


Woven Fabrics





$(E_1 = E_2)$



USES 2

Funded by the European Union

AIRBUS

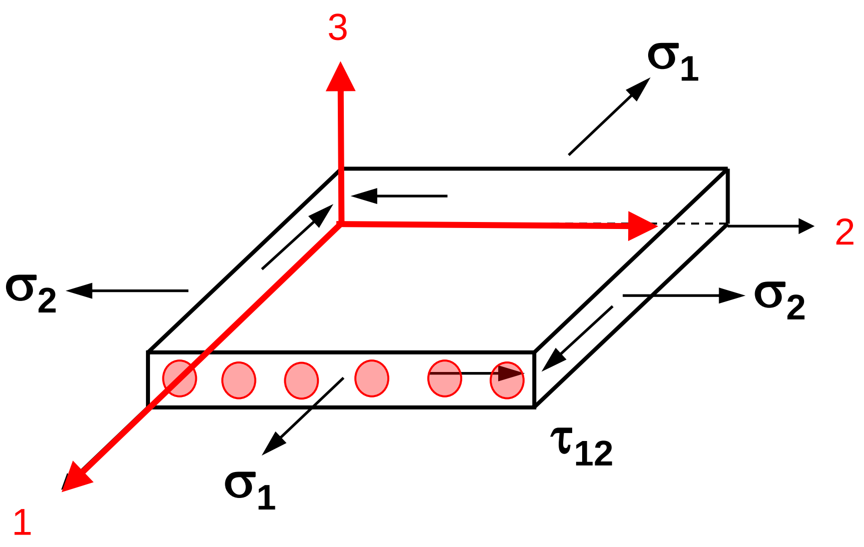
Introduction to mechanics of composite materials (ii)

Mechanics of the orthotropic ply

Ply properties and stress vs strain relationship

- ❑ (cured) ply thickness
  t_{ply} / cpt
- ❑ (in-plane) elastic moduli
  E_1, E_2, v_{12}, G_{12}
- ❑ (in-plane) ply strengths
  F_{1t}, F_{1c}, F_{2t}, F_{2c}, F_{12}
  X_t, X_c, Y_t, Y_c, S (*)

(*) typ. in American literature. Not generally recommended to adopt, as X, Y will be reserved for the laminate reference system in analysis



$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = [Q] \cdot \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix}$$

$$[S] = [Q]^{-1}$$

$$\begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix} = [S] \cdot \begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix}$$

$$[S] = \begin{bmatrix} \frac{1}{E_1} & \frac{-\nu_{21}}{E_2} & 0 \\ \frac{-\nu_{12}}{E_1} & \frac{1}{E_2} & 0 \\ 0 & 0 & \frac{1}{G_{12}} \end{bmatrix}$$

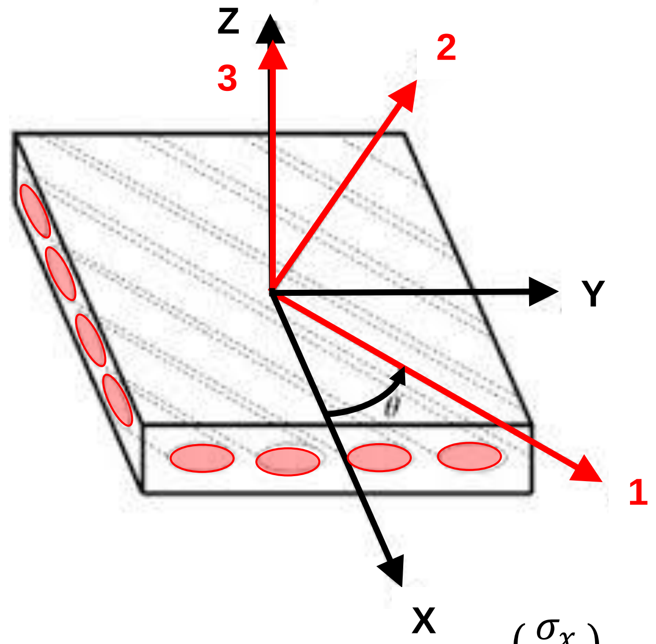
$$[Q] = \begin{bmatrix} \frac{E_1}{1-\nu_{12}.\nu_{21}} & \frac{E_1.\nu_{21}}{1-\nu_{12}.\nu_{21}} & 0 \\ \frac{E_2.\nu_{12}}{1-\nu_{12}.\nu_{21}} & \frac{E_2}{1-\nu_{12}.\nu_{21}} & 0 \\ 0 & 0 & G_{12} \end{bmatrix}$$

USES2 Funded by the European Union

Introduction to mechanics of composite materials (iii)

Mechanics of the orthotropic ply

Transformation of matrices to rotated reference system



$$\begin{bmatrix}\sigma_x\\ \sigma_y\\ \tau_{xy}\end{bmatrix}=\begin{bmatrix}\cos^2\theta & \sin^2\theta & -2\sin\theta\cos\theta\\ \sin^2\theta & \cos^2\theta & 2\sin\theta\cos\theta\\ \sin\theta\cos\theta & -\sin\theta\cos\theta & \cos^2\theta-\sin^2\theta\end{bmatrix}\begin{bmatrix}\sigma_1\\ \sigma_2\\ \tau_{12}\end{bmatrix}$$

$$\begin{bmatrix}\varepsilon_x\\ \varepsilon_y\\ \frac{\gamma_{xy}}{2}\end{bmatrix}=\begin{bmatrix}\cos^2\theta & \sin^2\theta & -2\sin\theta\cos\theta\\ \sin^2\theta & \cos^2\theta & 2\sin\theta\cos\theta\\ \sin\theta\cos\theta & -\sin\theta\cos\theta & \cos^2\theta-\sin^2\theta\end{bmatrix}\begin{bmatrix}\varepsilon_1\\ \varepsilon_2\\ \frac{\gamma_{12}}{2}\end{bmatrix}$$

$$[\bar{Q}]=[T]^{-1}[Q][T]^{-T}$$

$$\begin{bmatrix}\sigma_x\\ \sigma_y\\ \tau_{xy}\end{bmatrix}=[T]^{-1}\begin{bmatrix}\sigma_1\\ \sigma_2\\ \tau_{12}\end{bmatrix}$$

$$[T]=\begin{bmatrix}\cos^2\theta & \sin^2\theta & 2\sin\theta\cos\theta\\ \sin^2\theta & \cos^2\theta & -2\sin\theta\cos\theta\\ -\sin\theta\cos\theta & \sin\theta\cos\theta & \cos^2\theta-\sin^2\theta\end{bmatrix}$$

$$\begin{Bmatrix}\sigma_x\\ \sigma_y\\ \tau_{xy}\end{Bmatrix}=[\bar{Q}]\begin{Bmatrix}\varepsilon_x\\ \varepsilon_y\\ \gamma_{xy}\end{Bmatrix}$$

$$\bar{Q}_{11}=[Q_{11}.\cos^4\theta\,]+[2(Q_{12}+2Q_{66}).\cos^2\theta\,.\sin^2\theta]+[Q_{22}.\sin^4\theta]$$

$$\bar{Q}_{12}=[(Q_{11}+Q_{22}-4Q_{66}).\cos^2\theta\,.\sin^2\theta]+[Q_{12}.(\cos^4\theta+\sin^4\theta)]$$

$$\bar{Q}_{16}=[(Q_{11}-Q_{12}-2Q_{66}).sin\,\theta\,.\cos^3\theta]+[(Q_{12}-Q_{22}+2Q_{66}).\sin^3\theta\,.\cos\theta]$$

$$\bar{Q}_{22}=[Q_{11}.\sin^4\theta\,]+[2(Q_{12}+2Q_{66}).\cos^2\theta\,.\sin^2\theta]+[Q_{22}.\cos^4\theta]$$

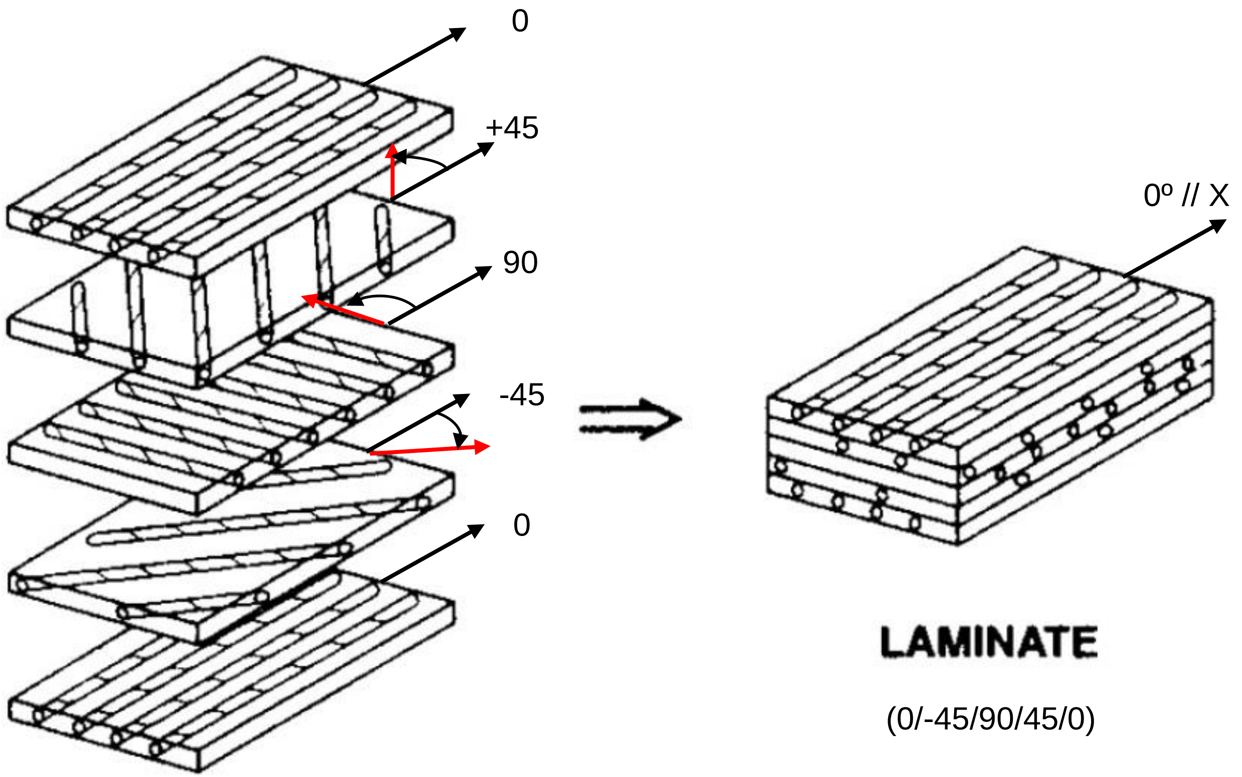
$$\bar{Q}_{26}=[(Q_{11}-Q_{12}-2Q_{66}).\sin^3\theta\,.\cos\theta]+[(Q_{12}-Q_{22}+2Q_{66}).sin\,\theta\,.\cos^3\theta]$$

$$\bar{Q}_{66}=[(Q_{11}+Q_{22}-2Q_{12}-2Q_{66}).\cos^2\theta\,.\sin^2\theta]+[Q_{66}.(\cos^4\theta+\sin^4\theta)]$$

Funded by the European Union

Introduction to mechanics of composite materials (iv)

The laminate ... as a stack of plies



(0/-45/90/45/0)

<u>Typ. conventions for stacking sequences</u>

- ✓ Symmetry with even number of plies:

  (0/0/-45/**45/45**/-45/0/0) = (0/0/-45/**45**)**S**

- ✓ Symmetry with odd number of plies:

  (0/0/-45/45/**90**/45/-45/0/0) = (0/0/-45/45/**90**)$

- ✓ Repetition of sub-laminate:

  (45/-45/90/0/45/-45/90/0) = $(45/-45/90/0)_2$

- ✓ Symmetry, sub-laminate repetition and other notations:

  $(45/-45/90/0)_2$S =

  = (45/-45/90/0/45/-45/90/0/0/90/-45/45/0/90/-45/45)

  $(+/-/90/0)_2$S = $(45/-45/90/0)_2$S

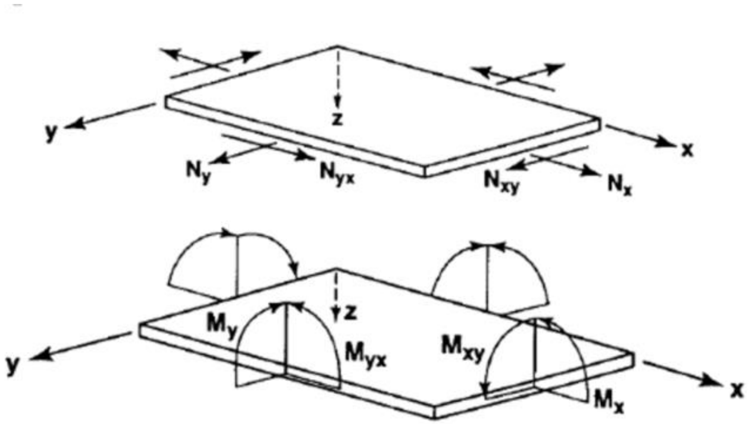
  ((+/-/90/0)S)S = (45/-45/90/0/0/90/-45/45)S

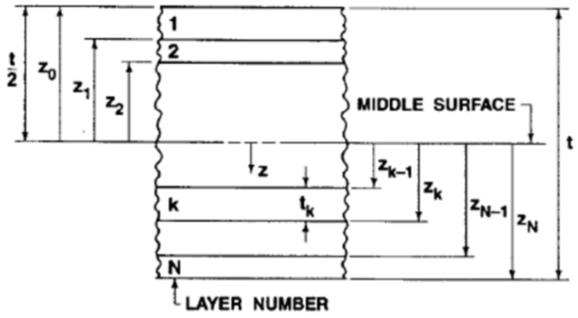
USES 2 Funded by the European Union

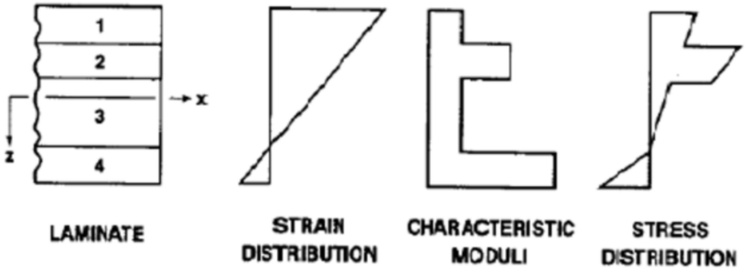
Introduction to mechanics of composite materials (v)

Classical Laminates Theory (CLT)

Classical laminated plate theory (a.k.a Classical Laminates Theory or Classical Laminates Analysis) reduces the stress/strain behavior of a structural laminate to a solvable two-dimensional mechanics of deformable bodies problem: Fundamental relationship between in-plane forces and bending moments (per unit length) in the laminated-plate, and the deformation of its mid-plane surface (extensional strains and curvatures)







$$\begin{bmatrix} N \\ M \end{bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix} \begin{bmatrix} \varepsilon \\ \kappa \end{bmatrix}$$

Funded by the European Union

Introduction to mechanics of composite materials (vi)

Laminate stiffness matrices

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \int_{z=-\frac{t}{2}}^{z=\frac{t}{2}} [\bar{Q}]_n . dz \begin{Bmatrix} \varepsilon_x{}^0 \\ \varepsilon_y{}^0 \\ \gamma_{xy}{}^0 \end{Bmatrix} + \int_{z=-\frac{t}{2}}^{z=\frac{t}{2}} [\bar{Q}]_n . z . dz \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix}$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \int_{z=-\frac{t}{2}}^{z=\frac{t}{2}} [\bar{Q}]_n . z . dz \begin{Bmatrix} \varepsilon_x{}^0 \\ \varepsilon_y{}^0 \\ \gamma_{xy}{}^0 \end{Bmatrix} + \int_{z=-\frac{t}{2}}^{z=\frac{t}{2}} [\bar{Q}]_n . z^2 . dz \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix}$$

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \\ M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix} . \begin{Bmatrix} \varepsilon_x{}^0 \\ \varepsilon_y{}^0 \\ \gamma_{xy}{}^0 \\ K_x \\ K_y \\ K_{xy} \end{Bmatrix}$$

Membrane stiffness

$$[A] = \int_{z=-\frac{t}{2}}^{z=\frac{t}{2}} [\bar{Q}]_n . dz = \sum_{n=1}^{nº\ plies} [\bar{Q}]_n . (z_{n2} - z_{n1}) = \sum_{n=1}^{nº\ plies} [\bar{Q}]_n . t_n$$

Bending/extension coupling

$$[B] = \int_{z=-\frac{t}{2}}^{z=\frac{t}{2}} [\bar{Q}]_n . z . dz = \sum_{n=1}^{nº plies} [\bar{Q}]_n . \frac{1}{2} \left(z_{n2}{}^2 - z_{n1}{}^2 \right) = \sum_{n=1}^{nº plies} [\bar{Q}]_n . t_n . z_n$$

Bending stiffness

$$[D] = \int_{z=-\frac{t}{2}}^{z=\frac{t}{2}} [\bar{Q}]_n . z^2 . dz = \sum_{n=1}^{nº\ plies} [\bar{Q}]_n . \frac{1}{3} \left(z_{n2}{}^3 - z_{n1}{}^3 \right) \cong \sum_{n=1}^{nº\ plies} [\bar{Q}]_n . t_n . z_n{}^2$$

Note:
Generalized/equivalent bending stiffness

Literature offers a “corrected” equivalent/generalized bending stiffness, if the laminate is only slightly unsymmetric/unbalanced, in order to deal with problems like buckling analysis, avoiding the complexity of coupled bending/membrane equations:

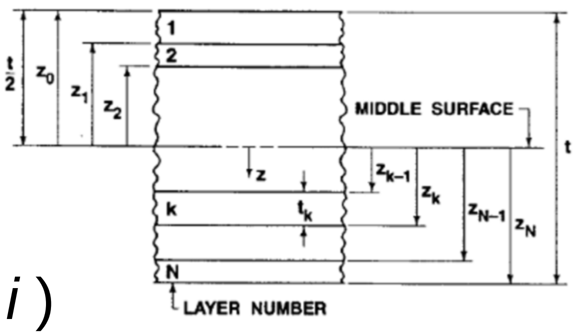
$$[D^*] = [D] - [B][A]^{-1}[B]$$

CMH-17, Vol. 3, formula 9.2.2.1(b)

USES 2 Funded by the European Union

Introduction to mechanics of composite materials (vii)

Ply strain/stress recovery



Mid-plane strain from the CLT equation *{NM} = [ABD] {εκ}*^{0}

$$\{\varepsilon\kappa\}^0 = \{ (\varepsilon_x, \varepsilon_y, \gamma_{xy})^0, (\kappa_x, \kappa_y, \kappa_{xy}) \}$$

Ply strain in laminate reference system (*z* that of the centre of ply *i*)

$$\{\varepsilon\}_i = \{\varepsilon\}^0 + z\cdot\{\kappa\} = (\varepsilon_x, \varepsilon_y, \gamma_{xy})_i$$

Ply strain in ply reference system

$$\begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix} = [T_\varepsilon] \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix}$$

$$[T_\varepsilon] = \begin{bmatrix} cos^2\,\theta & sin^2\,\theta & \sin\theta\cos\theta \\ sin^2\,\theta & cos^2\,\theta & -\sin\theta\cos\theta \\ -2\sin\theta\cos\theta & 2\sin\theta\cos\theta & cos^2\,\theta - sin^2\,\theta \end{bmatrix}$$

Ply stress in ply reference system

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = [Q] \,.\, \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix}$$

Funded by the European Union

Introduction to mechanics of composite materials (viii)

Typical analysis process

1) Stiffness matrices

   From ply properties (cpt , E_1 , E_2 , ν_{12} , G_{12}) and laminate stacking sequence, calculate the laminate stiffness matrices *[A], [B], [D]*

2) Mid-plane strain

   From laminate stiffness matrices and plate loads at the mid-plane of the laminate (N_x , N_y , N_{xy} , M_x , M_y , M_{xy}) calculate the mid-plane extensional (ε_x , ε_y , γ_{xy}) and curvature (κ_x , κ_y , κ_{xy}) strains, using the equation $\{NM\} = [ABD]\,\{\varepsilon\kappa\}^0$

3) Stress/strain in plies

   From the mid-plane strain, calculate the strain at each ply $(\varepsilon_x , \varepsilon_y , \gamma_{xy})_i$ and, using the material compliance matrix, and the applicable transformations at the ply angle, calculate the ply strain $(\varepsilon_1 , \varepsilon_2 , \gamma_{12})_i$ and stress $(\sigma_1 , \sigma_2 , \tau_{12})_i$ in its local reference axis system

USES 2 Funded by the European Union

Introduction to mechanics of composite materials (ix)

Simplified methods: apparent/equivalent elastic moduli

Skipping the matrix equations with quasi-orthotropic laminates under in-plane loads…

✓ For a laminate with a common symmetric and balanced stacking sequence

$$E_x = 1 / ([A^{-1}]_{11} \cdot t) = (A_{11} – A_{12}^2 / A_{22}) / t$$
$$E_y = 1 / ([A^{-1}]_{22} \cdot t) = (A_{22} – A_{12}^2 / A_{11}) / t$$
$$\nu_{xy} = - E_x \cdot t \cdot [A^{-1}]_{12} = A_{12} / A_{22}$$
$$\nu_{xy} / E_x = \nu_{yx} / E_y \quad \rightarrow \quad \nu_{yx} = \nu_{xy} \cdot E_y / E_x = A_{12} / A_{11}$$
$$G_{xy} = 1 / ([A^{-1}]_{66} \cdot t) = A_{33} / t$$

✓ Laminate elastic moduli and in-plane strain in quasi-orthotropic laminates

$$\varepsilon_{xx} = N_x / (E_x . t) - N_y \cdot \nu_{xy} / (E_x . t)$$
$$\varepsilon_{yy} = N_y / (E_y . t) - N_x \cdot \nu_{yx} / (E_y . t)$$
$$\gamma_{xy} = N_{xy} / (G_{xy} . t)$$

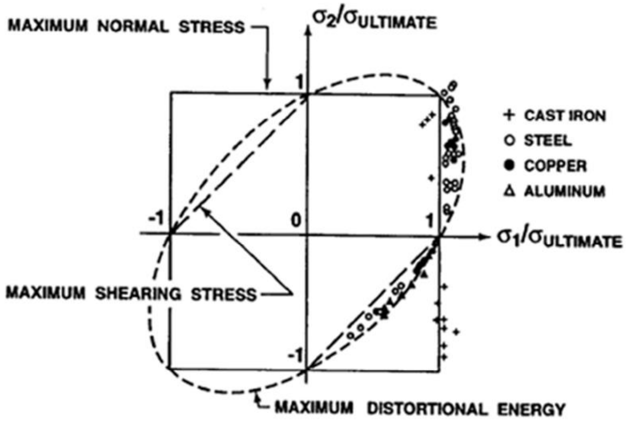
$$\varepsilon_{45} = (\varepsilon_{xx} + \varepsilon_{yy} + \gamma_{xy}) / 2$$
$$\varepsilon_{-45} = (\varepsilon_{xx} + \varepsilon_{yy} - \gamma_{xy}) / 2$$

USES 2 Funded by the European Union

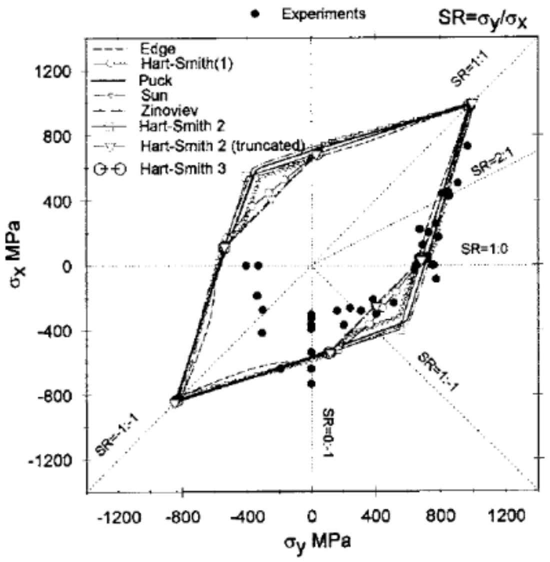
Failure of the fibrous composite material (i)

Introduction: limited failure prediction capability

“Current commercial design practices place little or no reliance on the ability to predict the ultimate strength of the structure with any great accuracy. Failure theories are often used in the initial calculations to 'size' a component (i.e. to establish the approximate dimensions, such as panel thickness, width, etc). Beyond that point, experimental tests on coupons or structural elements (such as the notched hot/wet compressive strength tests in aerospace) are used to determine the global design allowables.” (1st WWFE)



Metals

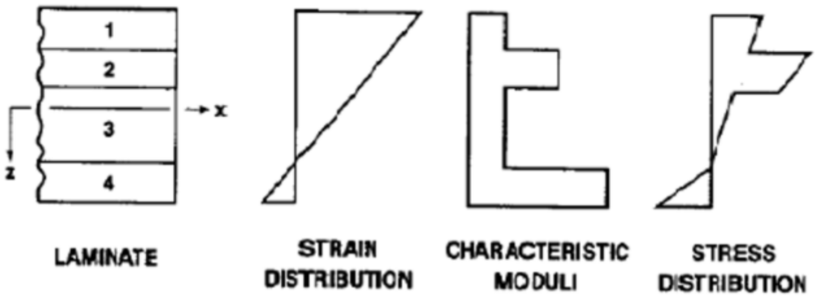


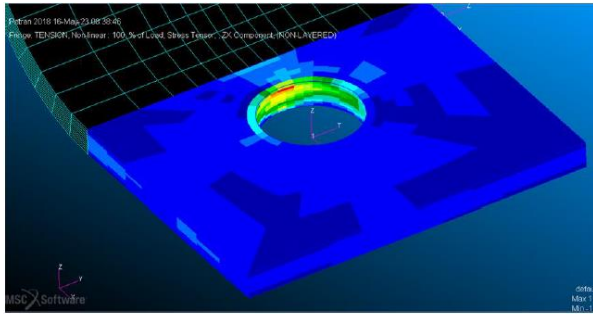
(0°/±45°/90°) AS4/3501-6 laminates (Test Case No 6).

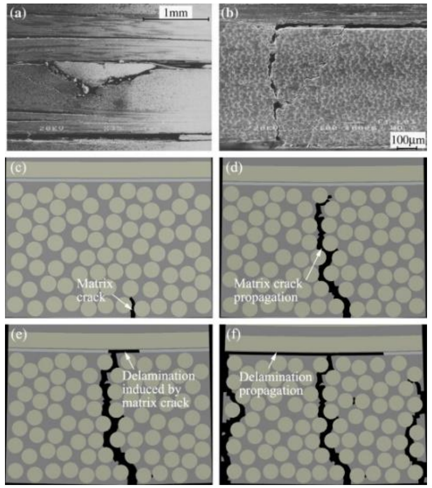
Failure of the fibrous composite material (ii)

Introduction: scales

- Macroscopic scale: the failure is calculated at the scale of the laminate
- Mesoscopic scale: The determination of the failure of the laminate is determined at the scale of the ply. The level of stress (or strain) in the plies allows determining the failure of the laminate.
- Microscopic scale: the failure of the laminate is calculated at the scale of the constituents of the ply (fibre and resin). This type of analysis takes into account the interaction between resin and fibres in order to determine the behavior of a ply.



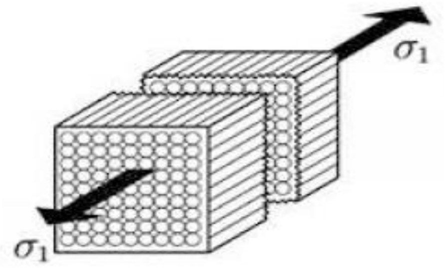




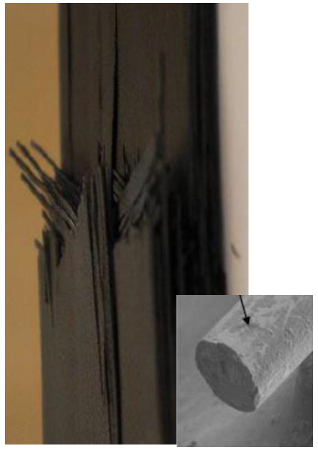
Funded by the European Union

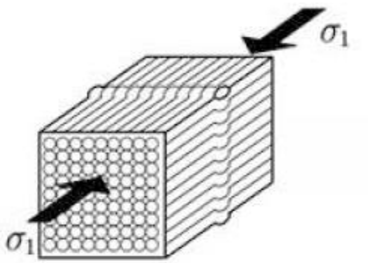
Failure of the fibrous composite material (iii)

Failure mechanisms: intra-ply…

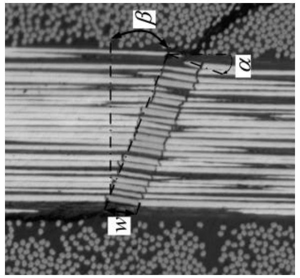


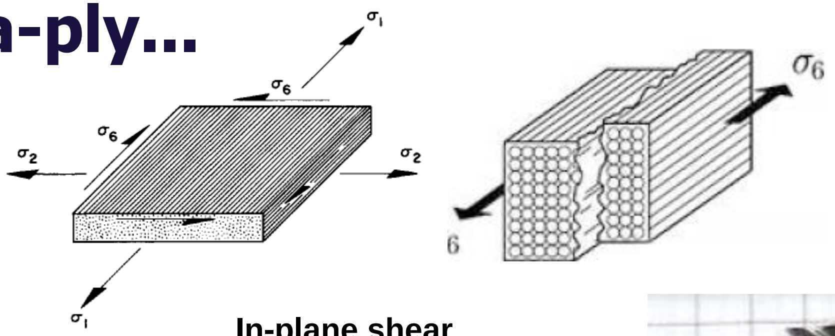
Longitudinal tension
Fiber failure, Fiber pull-out/debonding from matrix, ….



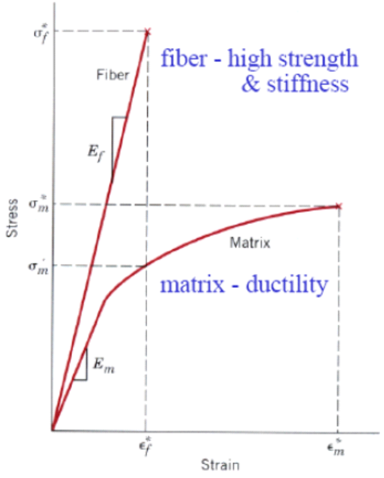


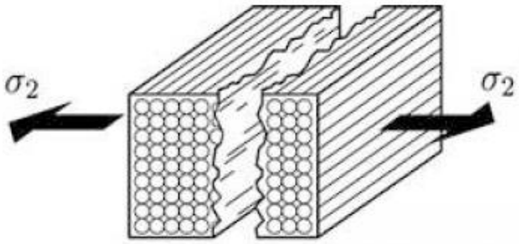
Longitudinal compression
Fiber “kinking” (micro-buckling), Matrix cracking, ….



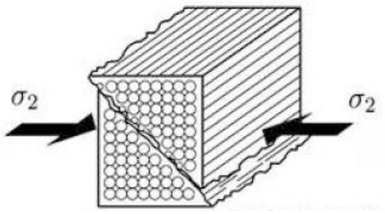


In-plane shear
Matrix (shear) failure, ….





Transverse tension
Matrix (tension) failure with stress concentrations around fibers, ….

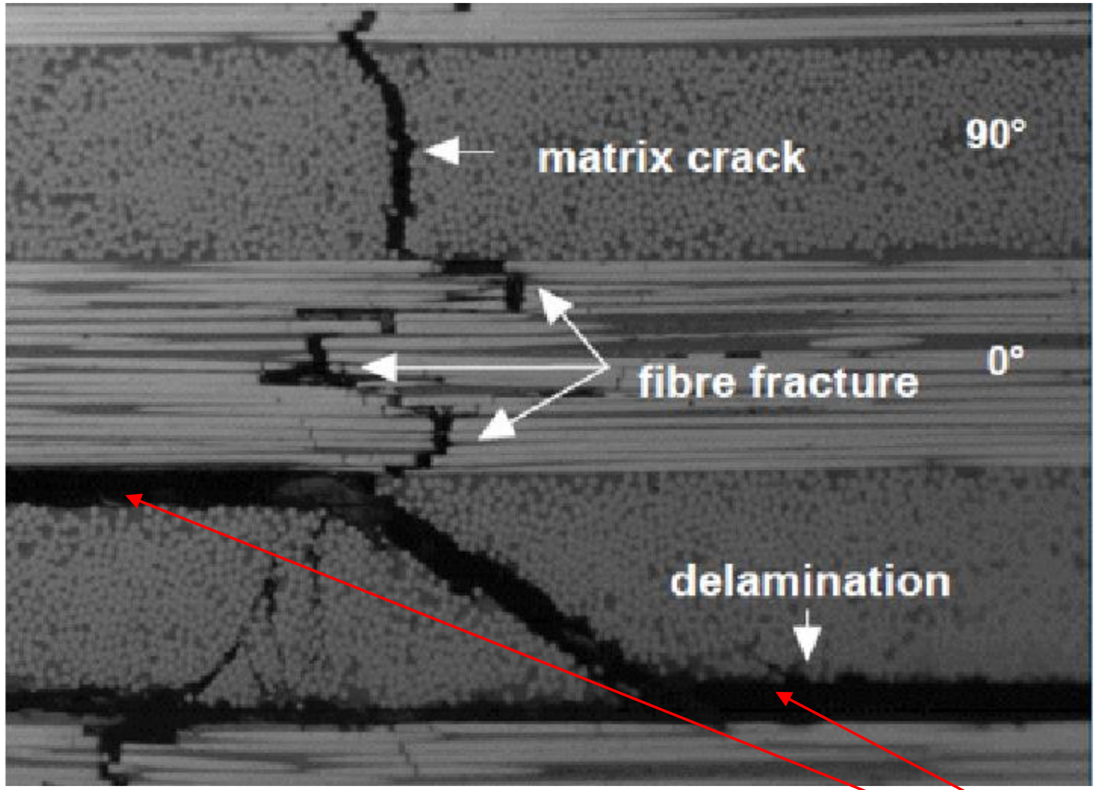


Transverse compression
Matrix (shear) failure, ….

USES 2
Funded by the European Union

Failure of the fibrous composite material (iv)

… and delamination



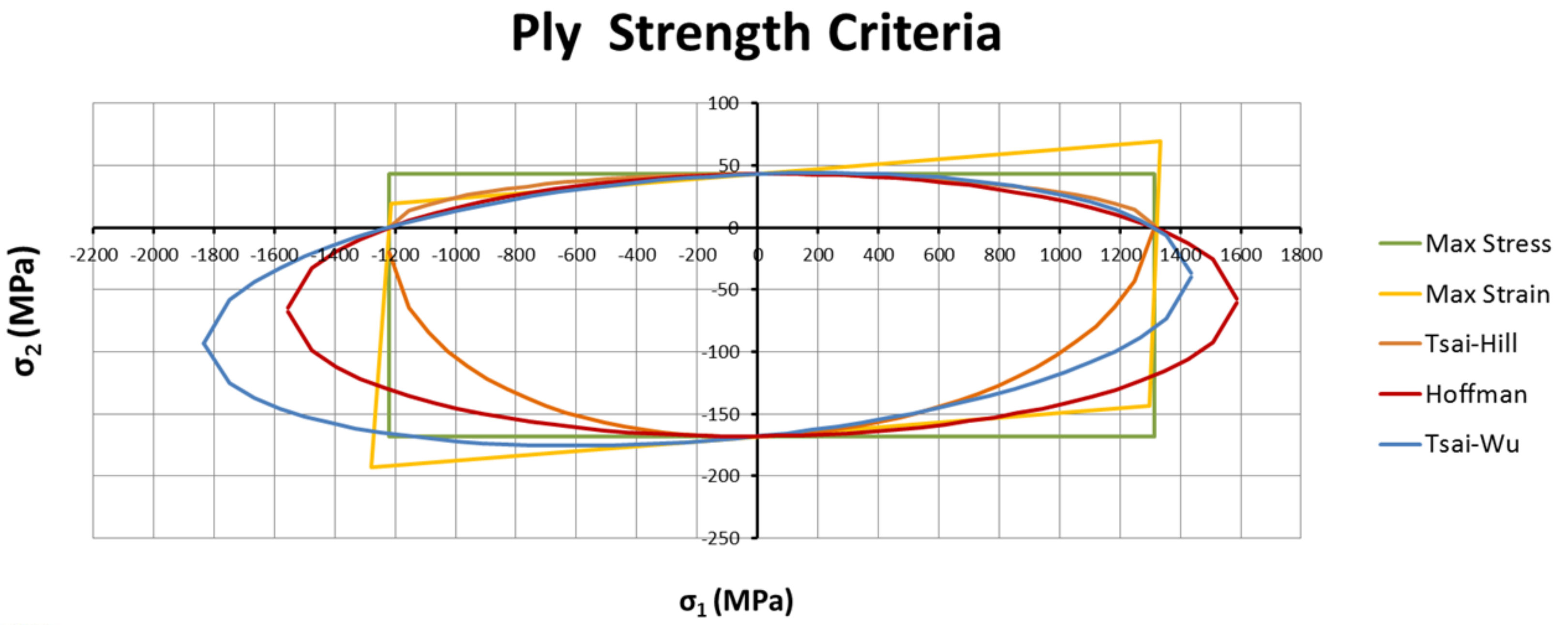
and delaminations!
(laminate level)

Failure of the fibrous composite material (v)

Overview of failure theories for the orthotropic composite ply

Regarding treatment of stress components
- Non-interactive, e.g. max. stress, max. strain
- Interactive, eg. fully interactive theories of Hoffman, Tsai-Hill, Tsai-Wu

Regarding consideration of failure mechanisms
- Differentiating or not failure mechanisms : failure of fiber, of matrix, debonding… (up to some extent), e.g. Puck, LaRC03



Failure of the fibrous composite material (vi)

Some classical failure theories

1. Independent conditions	(a) Maximum stress	$\sigma_1 = X_t$ or $-X_c$ or $\sigma_2 = Y_t$ or $-Y_c$ or $\lvert\sigma_6\rvert = Q$
	(b) Maximum strain	$\varepsilon_1 = \varepsilon_{Xt}$ or $-\varepsilon_{Xc}$ or $\varepsilon_2 = \varepsilon_{Yt}$ or $-\varepsilon_{Yc}$ or $\lvert\varepsilon_6\rvert = \varepsilon_Q$

Extract of ESDU 83014 (1983)

2. Independent – partly interactive	(a) Grant-Sanders	$\sigma_1 = X_t$ or $-X_c$ or ε_{m1} or $\varepsilon_{m2} = \varepsilon_{mc}$ or $\lvert\sigma_6\rvert = Q_f$ or Q_m	plus tension/shear and compression/shear interaction formulae
	(b) Puck	$\sigma_1 = X_t$ or $-X_c$ or $\left(\frac{\sigma_2}{Y}\right)^2 + \left(\frac{\sigma_6}{Q}\right)^2 = 1$	
	(c) Puck modified	$\sigma_1 = X_t$ or $-X_c$ or $\frac{\sigma_2^2}{Y_t Y_c} + \sigma_2\left(\frac{1}{Y_t} - \frac{1}{Y_c}\right) + \left(\frac{\sigma_6}{Q}\right)^2 = 1$	

3. Fully interactive	(a) Tsai-Hill	$\left(\frac{\sigma_1}{X}\right)^2 - \frac{\sigma_1\sigma_2}{X^2} + \left(\frac{\sigma_2}{Y}\right)^2 + \left(\frac{\sigma_6}{Q}\right)^2 = 1$
	(b) Tsai-Wu	$A_{11}\sigma_1^2 + 2A_{12}\sigma_1\sigma_2 + A_{22}\sigma_2^2 + A_{66}\sigma_6^2 + B_1\sigma_1 + B_2\sigma_2 = 1$
	(c) Puppo-Evensen	$\left(\frac{\sigma_1}{X}\right)^2 - \phi\frac{X}{Y}\frac{\sigma_1}{X}\frac{\sigma_2}{Y} + \phi\left(\frac{\sigma_2}{Y}\right)^2 + \left(\frac{\sigma_6}{Q}\right)^2 = 1$ or $\phi\left(\frac{\sigma_1}{X}\right)^2 - \phi\frac{Y}{X}\frac{\sigma_1}{X}\frac{\sigma_2}{Y} + \left(\frac{\sigma_2}{Y}\right)^2 + \left(\frac{\sigma_6}{Q}\right)^2 = 1$

Hoffman

$$-\frac{\sigma_1^2}{X_cX_t} + \frac{\sigma_1\sigma_2}{X_cX_t} - \frac{\sigma_2^2}{Y_cY_t} + \frac{X_c+X_t}{X_cX_t}\sigma_1 + \frac{Y_c+Y_t}{Y_cY_t}\sigma_2 + \frac{\tau_{12}^2}{S_{12}^2} = 1$$

Yamada and Sun Criterion (1978)

$$\left(\frac{\sigma_{11}}{X}\right)^2 + \left(\frac{\sigma_{12}}{S_{is}}\right)^2 = 1$$

Hashin Criterion 2D (1980)

Tensile fiber mode $\sigma_{11} > 0$

$$\left(\frac{\sigma_{11}}{X_T}\right)^2 + \left(\frac{\sigma_{12}}{S}\right)^2 = 1$$

Compressive fiber mode $\sigma_{11} < 0$

$$\lvert\sigma_{11}\rvert = X_C$$

Tensile matrix mode $(\sigma_{22} > 0)$

$$\left(\frac{\sigma_{22}}{Y_T}\right)^2 + \left(\frac{\sigma_{12}}{S}\right)^2 = 1$$

Compressive matrix mode $(\sigma_{22} < 0)$

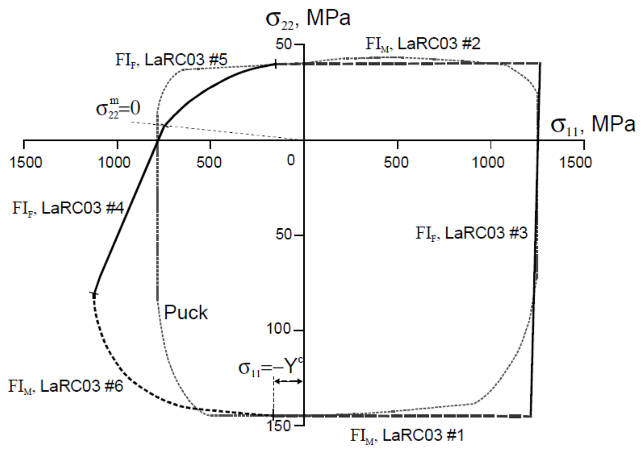
$$\left(\frac{\sigma_{22}}{2S_T}\right)^2 + \left[\left(\frac{Y_C}{2S_T}\right)^2 - 1\right]\frac{\sigma_{22}}{Y_C} + \left(\frac{\sigma_{12}}{S}\right)^2 = 1$$

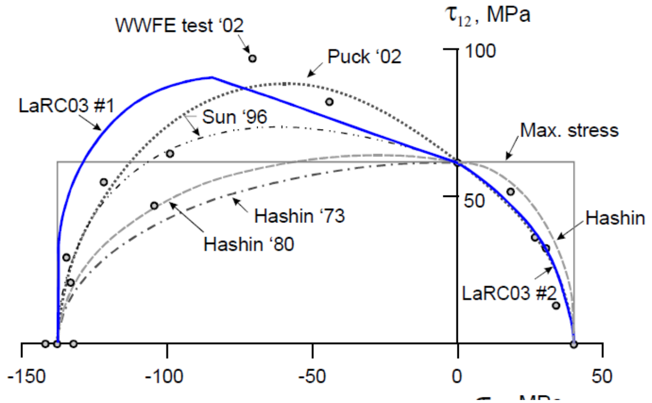
Failure of the fibrous composite material (vii)

LaRC series failure criteria

Appendix. Summary of LaRC03 Failure Criteria

	Matrix tension, $\sigma_{22} \geq 0$	Matrix compression, $\sigma_{22} < 0$	
Matrix Cracking		$\sigma_{11} < Y^C$	$\sigma_{11} \geq Y^C$
	$FI_M = (1-g)\left(\frac{\sigma_{22}}{Y_{is}^T}\right) + g\left(\frac{\sigma_{22}}{Y_{is}^T}\right)^2 + \left(\frac{\tau_{12}}{S_{is}^L}\right)^2$	$FI_M = \left(\frac{\tau_{eff}^{mT}}{S^T}\right)^2 + \left(\frac{\tau_{eff}^{mL}}{S_{is}^L}\right)^2$	$FI_M = \left(\frac{\tau_{eff}^{T}}{S^T}\right)^2 + \left(\frac{\tau_{eff}^{L}}{S_{is}^L}\right)^2$
	Fiber tension, $\sigma_{11} \geq 0$	Fiber compression, $\sigma_{11} < 0$	
Fiber Failure		$\sigma_{22}^m < 0$	$\sigma_{22}^m \geq 0$
	$FI_F = \frac{\varepsilon_{11}}{\varepsilon_1^T}$	$FI_F = \left\langle \frac{\lvert\tau_{12}^m\rvert + \eta^L \sigma_{22}^m}{S_{is}^L} \right\rangle$	$FI_F = (1-g)\left(\frac{\sigma_{22}^m}{Y_{is}^T}\right) + g\left(\frac{\sigma_{22}^m}{Y_{is}^T}\right)^2 + \left(\frac{\tau_{12}^m}{S_{is}^L}\right)^2$
Required Unidirectional Material Properties: $E_1, E_2, G_{12}, \upsilon_{12}, X^T, X^C, Y^T, Y^C, S^L, G_{Ic}(L), G_{IIc}(L)$			
Optional Properties: α_0, η^L			





NASA/TM-2003-212663 "Failure Criteria for FRP Laminates in Plane Stress", 2003

Carlos G. Dávila Langley Research Center, Hampton, Virginia
Pedro P. Camanho University of Porto, Porto, Portugal

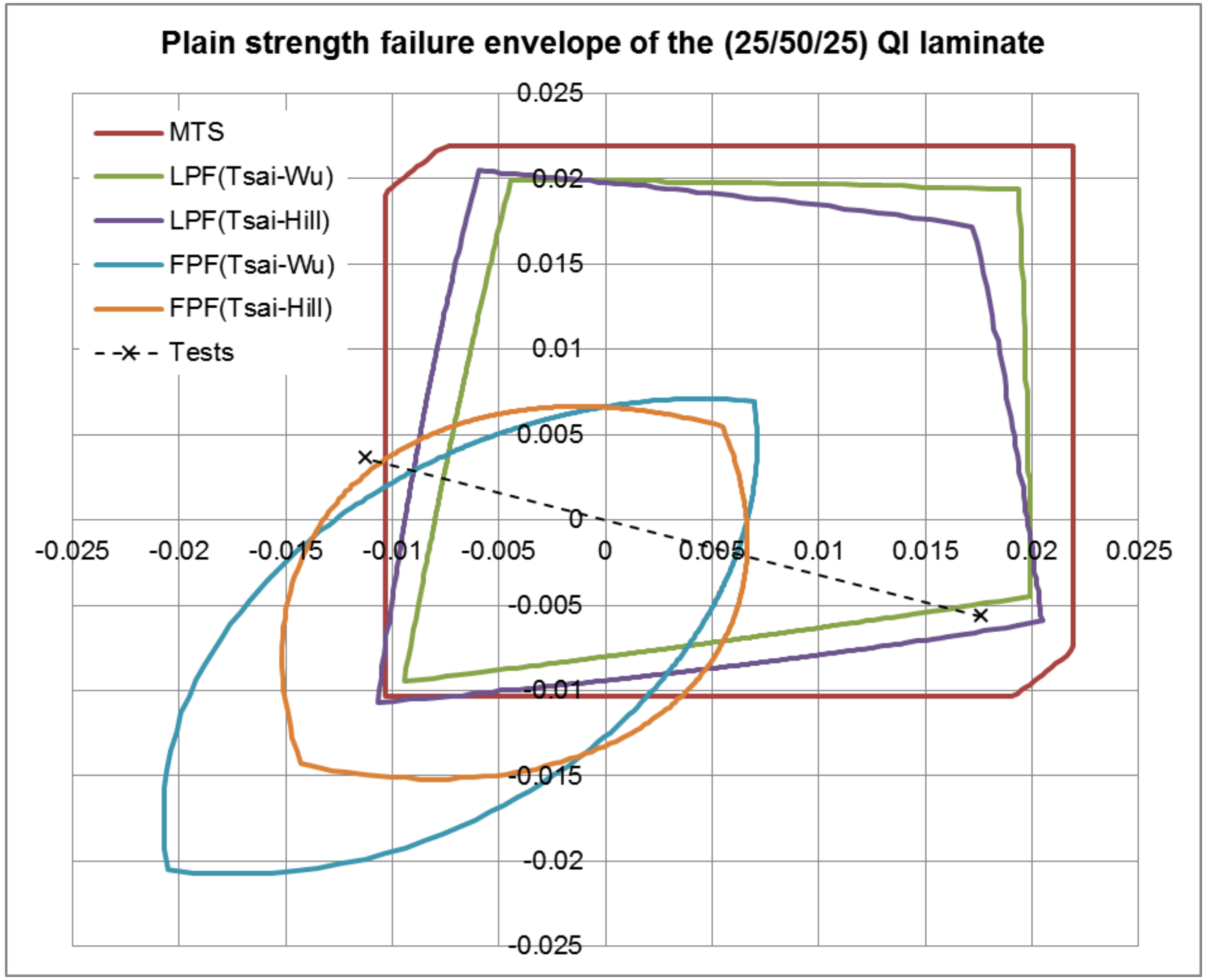
Funded by the European Union

Laminate failure (plain strength) (i)

First ply failure is laminate failure?

Assuming the failure of the first ply implies the catastrophic failure of the laminate will lead to overly-conservative results

Accepting that the failure of a ply, in particular if the failure mode affects the matrix and not the reinforcement fibres, does not imply a catastrophic failure, is an accepted approach of methods intending to predict laminate strength…



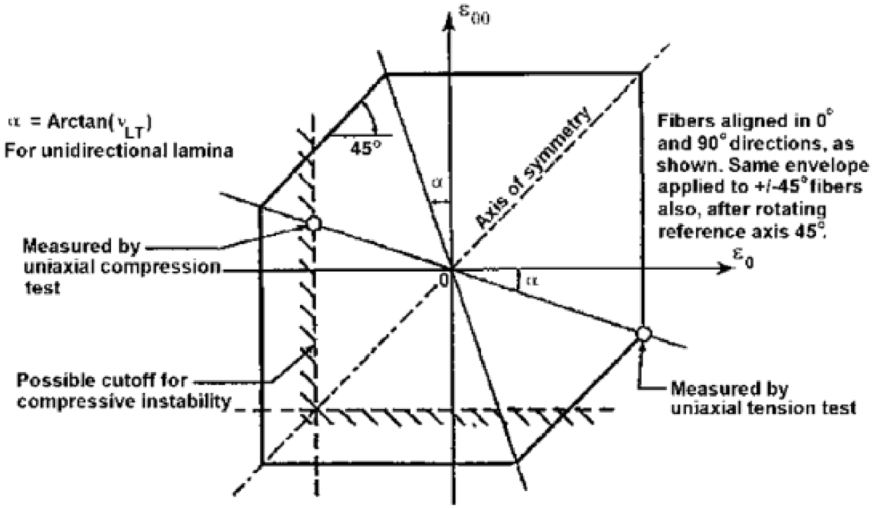
USES 2
Funded by the European Union

AIRBUS

Laminate failure (plain strength) (ii)

Maximum Truncated Strain

The Maximum Truncated Strain Failure Criterion, by L. J. Hart-Smith, is recommended by the CMH-17 for laminates acc. conventional design rules

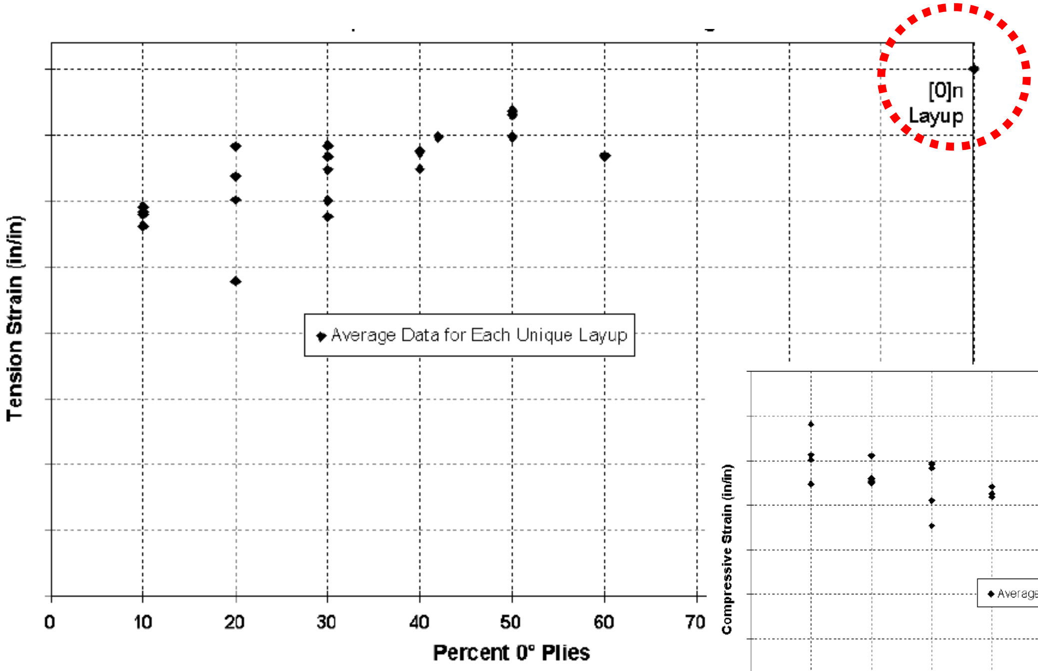


$$\varepsilon_{11}^{cu} \le \varepsilon_{11}^{i} \le \varepsilon_{11}^{tu}$$

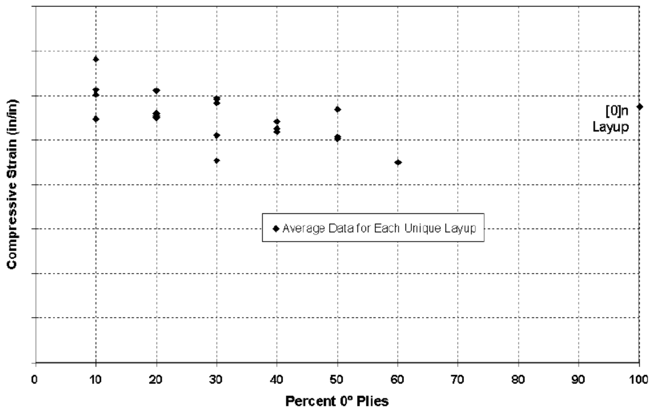
$$\varepsilon_{11}^{cu} \le \varepsilon_{22}^{i} \le \varepsilon_{11}^{tu}$$

$$|\varepsilon_{11}^{i} - \varepsilon_{22}^{i}| \le \left(1 + \nu_{LT}^{lamina}\right) |\varepsilon_{11}^{tu} \text{ or } \varepsilon_{11}^{cu}|^{*}$$

* whichever is greater

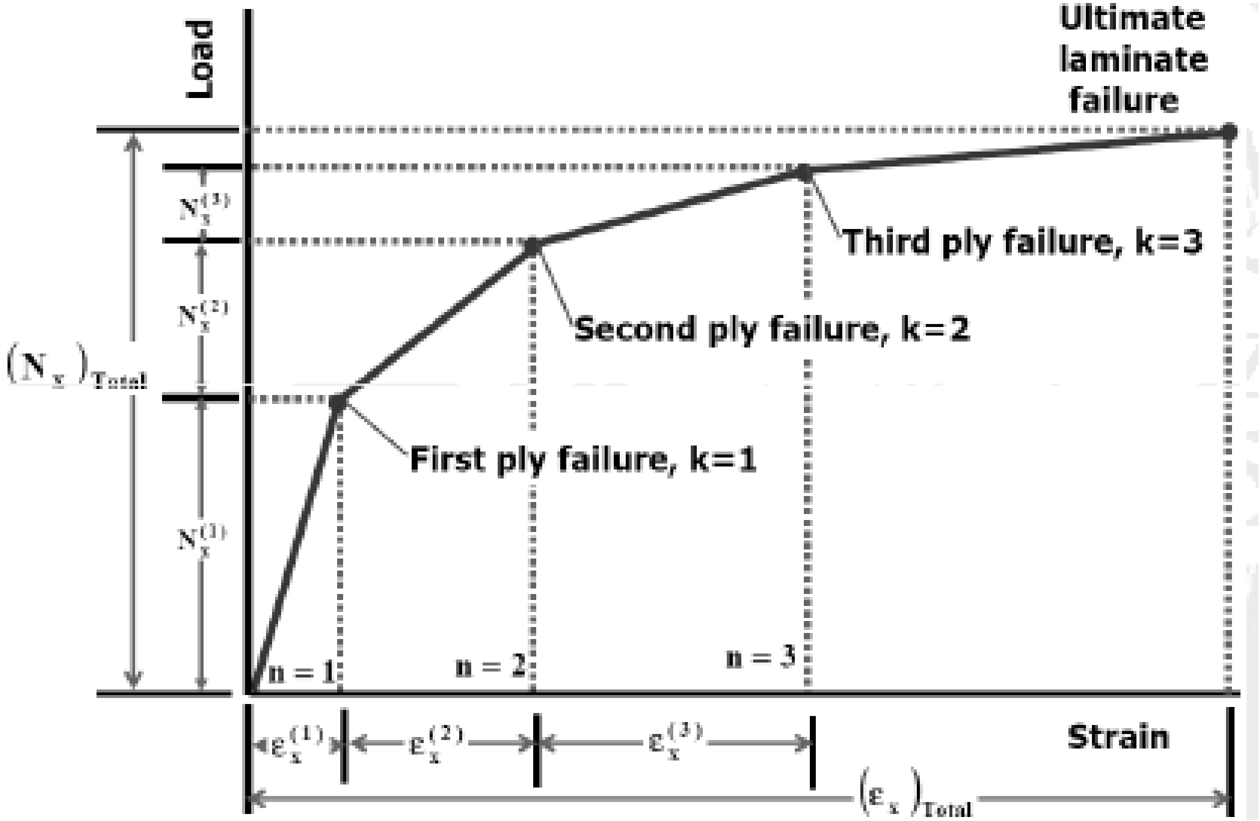


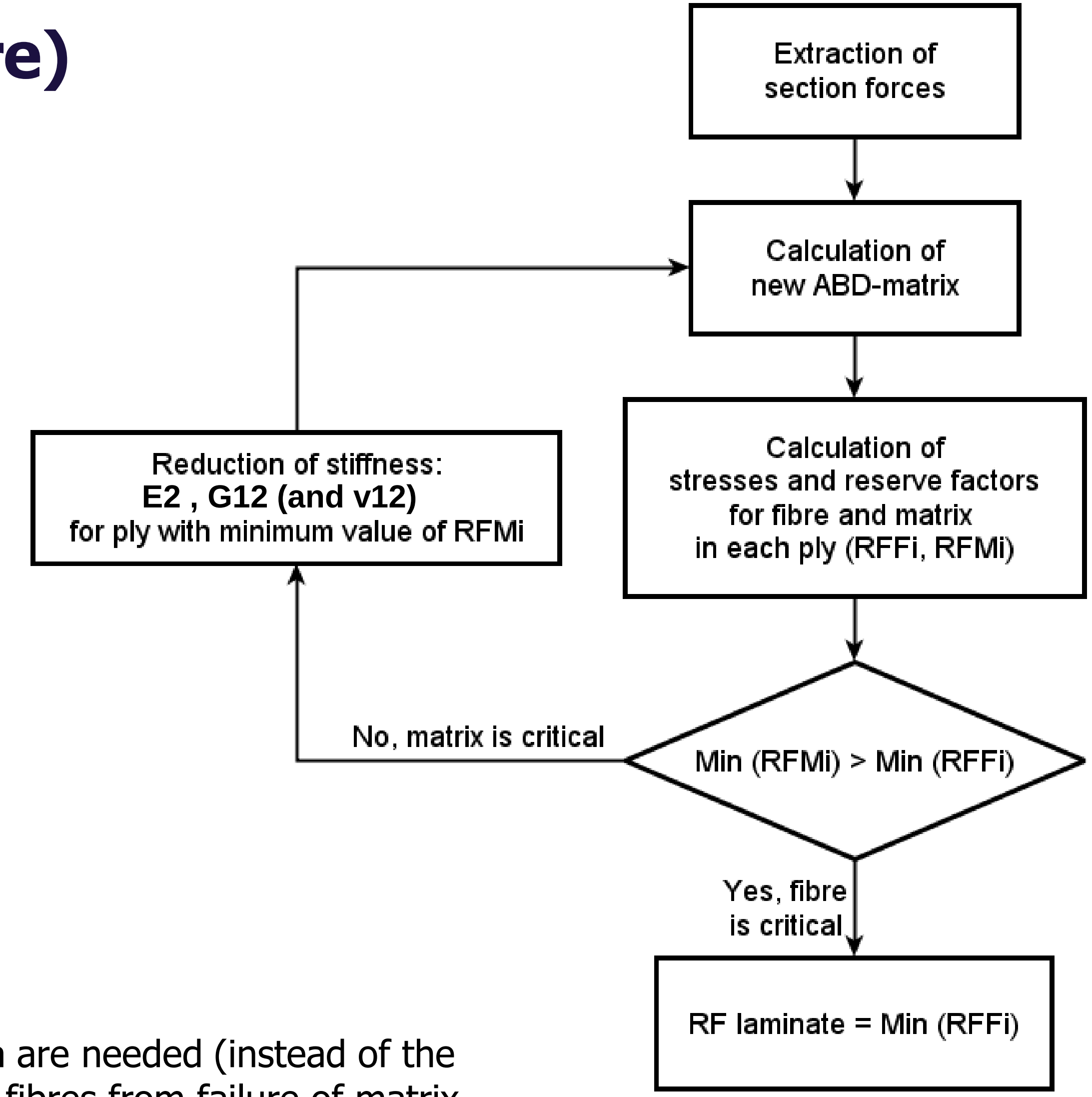
Using $\varepsilon_{1t} = F_{1t}/E_1$ unconservative for most laminates



Laminate failure (plain strength) (iii)

Progressive failure (First Fibre Failure)





Remark: Phenomenological failure criteria are needed (instead of the fully interactive), differentiating failure of fibres from failure of matrix.

USES 2
Funded by the European Union

Laminate failure (plain strength) (iv)

Simultaneous degradation of matrix

“Progressive degradation is an iterative, time-consuming process. Simultaneous degradation is a simplified approach in which all plies are degraded after the FPF is reached. The ultimate stress is reached by the ply having the lowest strength ratio among all degraded plies. This simultaneous degradation scheme uses only matrix degradation.” (Tsai)

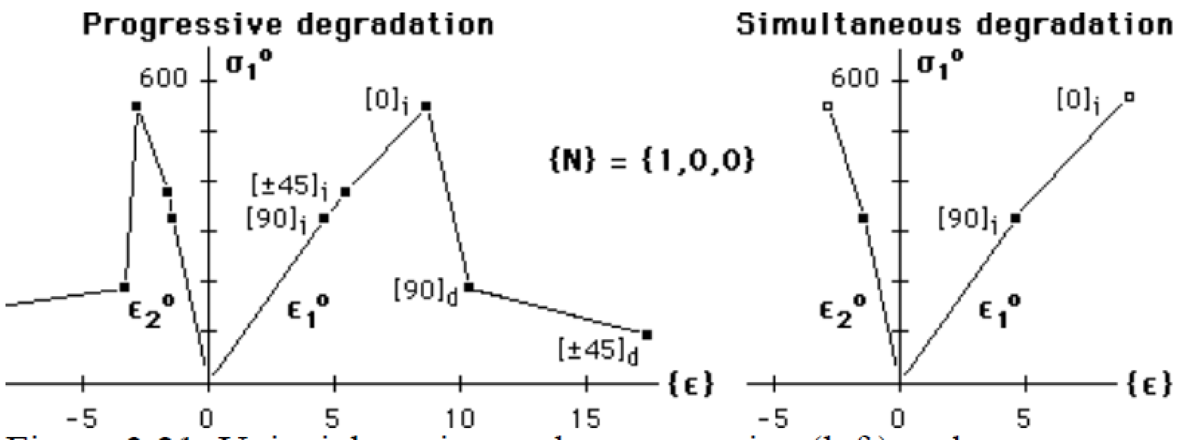


Figure 2.21. Uniaxial tension under progressive (left) and simultaneous (right) degradations for [π/4] of T300/5208.

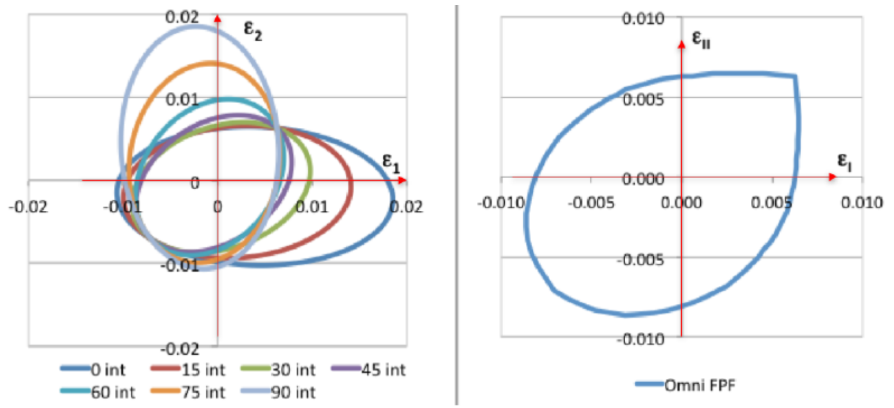
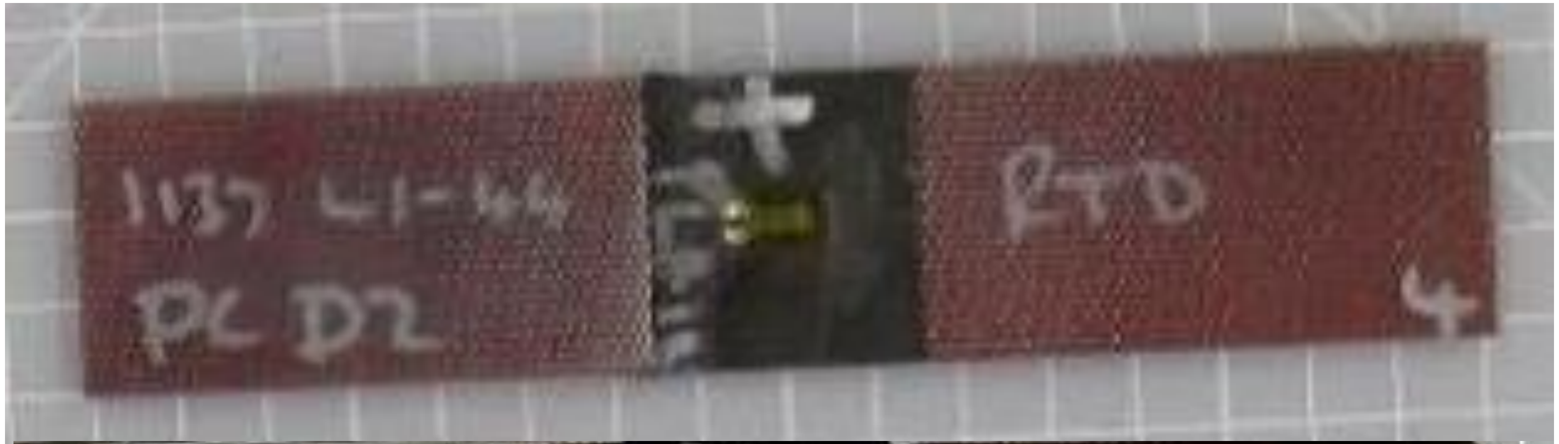


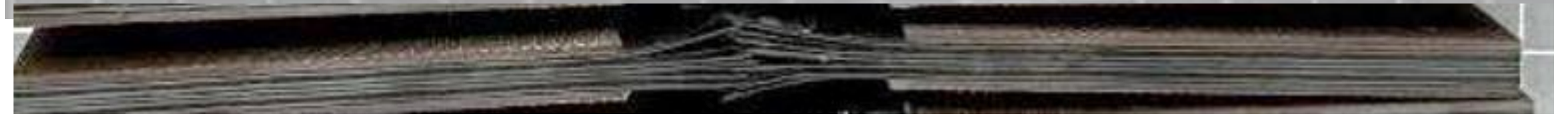
Figure 2.25. Generation of omni strain FPF envelope for IM7/977-3 using Tsai-Wu with $F_{xy}^*= -1/2$.

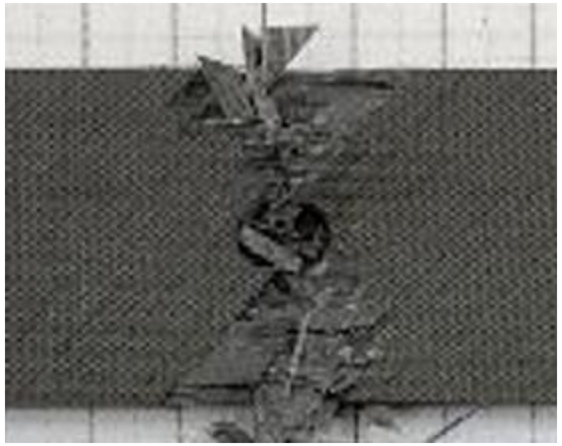
USES 2
Funded by the European Union
AIRBUS

Laminate failure (plain strength) (v)

Failure of multi-angular laminates in tension and compression







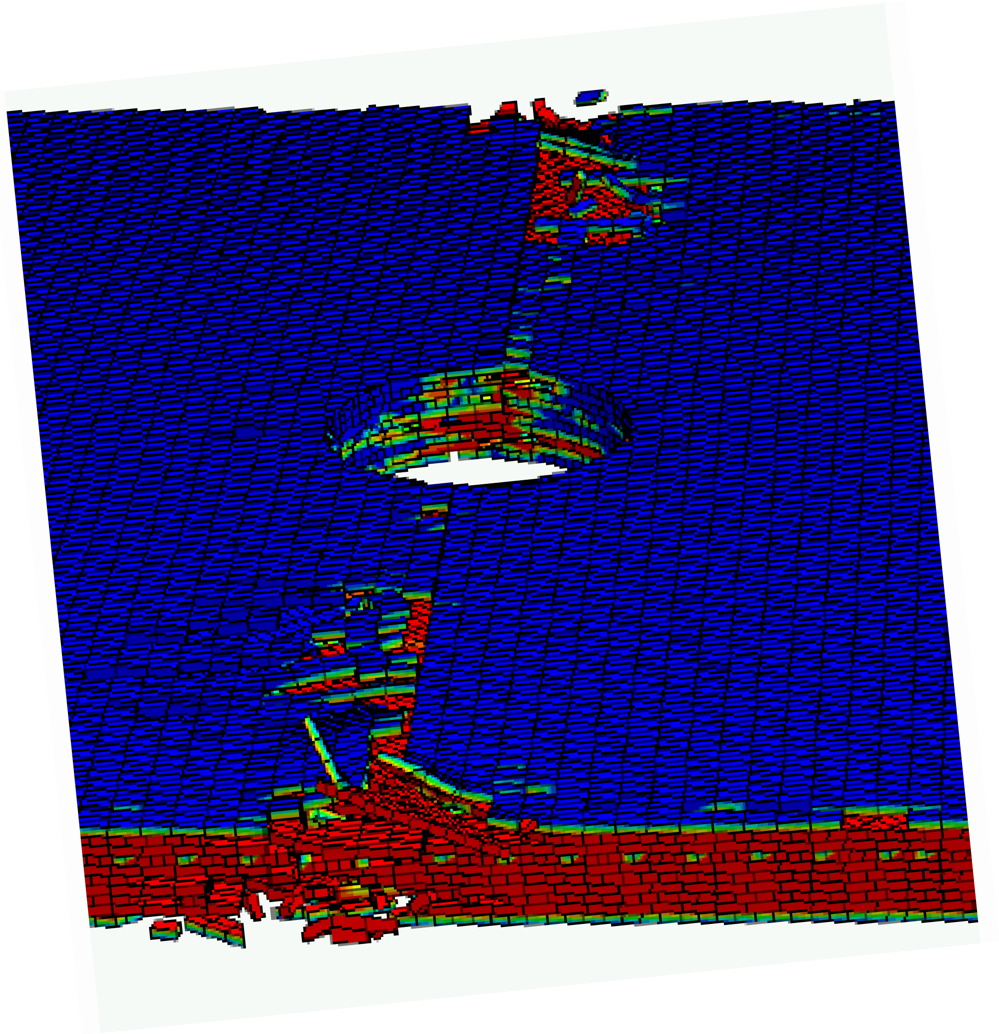
Funded by the European Union

PDFA in advanced numerical simulation (i)

PDFA and Finite Elements Models

Join capabilities of the FE analysis of calculate precisely stress states in structures with PDFA (Progressive Damage and Failure Analysis) methodologies to predict with accuracy failure in composite structures

"Many modes of damage can be observed in composite materials, including matrix cracks, fiber breakage, fiber-matrix de-bonding, and so on. ... Continuum Damage Mechanics (CDM) represents all these failure modes by the effect they have on the mesoscale behavior (lamina level) of the material. That is, CDM calculates the degraded moduli of the laminas and laminate in terms of continuum damage variables. Then, either strength or fracture mechanics failure criteria are used to detect damage initiation" (E. J. Barbero)



PDFA in advanced numerical simulation (ii)

Damage initiation

Before applicable failure criterion detect a point in the material constituents (fibre, material, ...) the material is undamaged *(D=0)*. Modern implementations use state of the art failure criteria (eg. LaRC series) and model non-linear behaviors of the constituents (matrix).

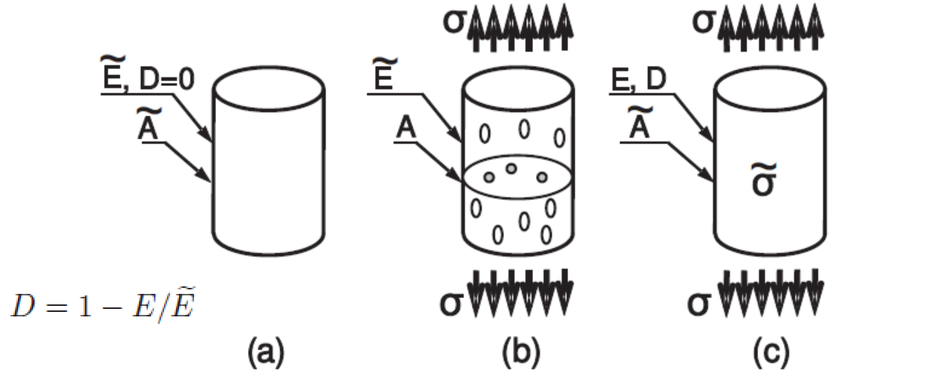


Fig. 8.1: (a) Unstressed material configuration, (b) stressed material configuration with distributed damage, (c) effective configuration.

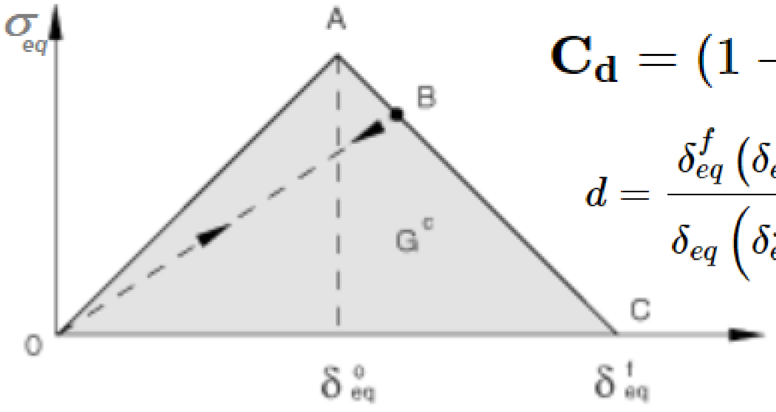
Damage evolution

After first failure (beyond A) the material is damaged *(D>0)*. Damage variable are computed from stress/strain levels and material constitutive laws (fracture energies playing a major role with ultimate stresses and strains), expressed as stress-displacement relationship. Material stiffness *(C_d)* is degraded, etc...

$$\sigma = \mathbf{C_d}\varepsilon^{el}$$

$$\mathbf{C_d} = (1 - D)\,\mathbf{C}$$

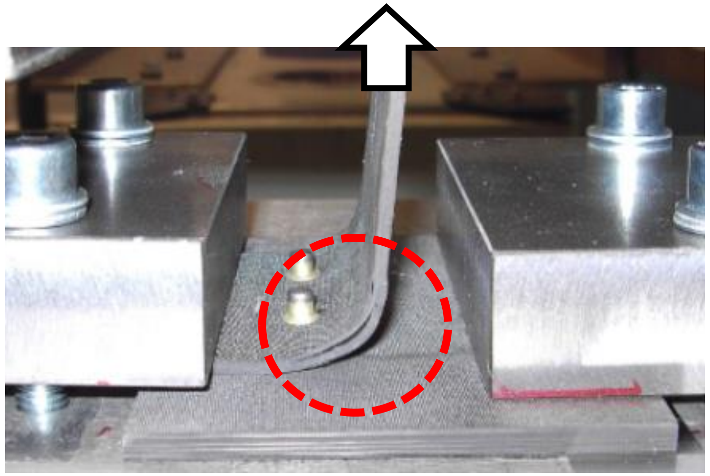
$$d = \frac{\delta_{eq}^{f}\left(\delta_{eq} - \delta_{eq}^{0}\right)}{\delta_{eq}\left(\delta_{eq}^{f} - \delta_{eq}^{0}\right)}$$

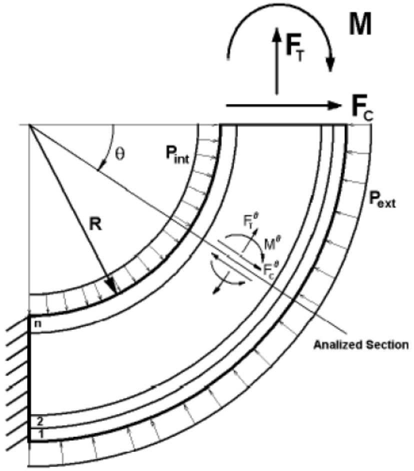


Delamination and bonded joints (i)

Through-thickness direct stresses and edge effects

- Unfolding in composite angles





Calculation of stresses in plies and their interfaces along the curved beam using, e.g. Lekhnitskii's 2D approach (Stress function formulation), detailed FEM analysis (*), ...

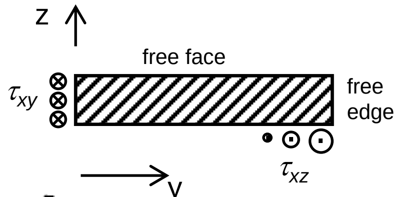
(*) Using FE analysis, can be combined with CDM methods to predict failure by PDFA , e.g. with cohesive material models / elements

Failure criterion for delamination

$$RF = \frac{1}{\sqrt{\left(\frac{\sigma_r}{f_{rr}^t}\right)^2 + \left(\frac{\tau_{r\theta}}{f_{r\theta}}\right)^2}} \quad \text{if } \sigma_r > 0$$

$$RF = \frac{f_{r\theta}}{\tau_{r\theta}} \quad \text{if } \sigma_r < 0$$

- Edge effects



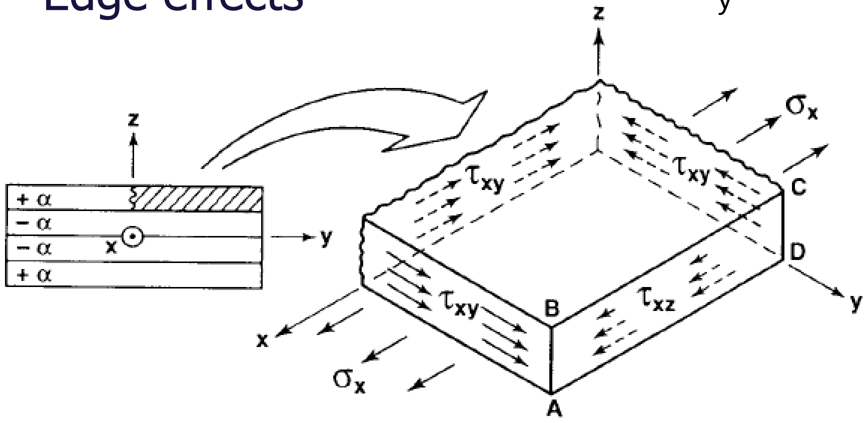


Figure 4-51 Free-Body Diagram for an Angle-Ply Laminate

Alternatively, consider the free-body diagram of half of the top layer of the four-layered angle-ply laminate in Figure 4-51. There, the left-hand side in the x-z plane is far from a free edge, so can have τ_{xy} as predicted with classical lamination theory. In contrast, at the free edge, as in Figure 4-51, τ_{xy} cannot exist on face ABCD. That is, ABCD must be stress-free because it is a free edge. Moreover, τ_{xy} on the front and back faces must go to zero at AB and CD. To achieve force equilibrium in the x-direction, we must identify a stress that could replace the action of the stress τ_{xy} that cannot exist on face ABCD. The only possible such stress is τ_{xz} that must exist on the bottom of the top-layer free-body diagram. For moment equilibrium about a vertical axis, τ_{xz} must be quite high because it exists only near the free edge. Although we know the stress (τ_{xz}) we are looking for and that it is high, we cannot determine how high without appealing to elasticity theory in the next subsection.

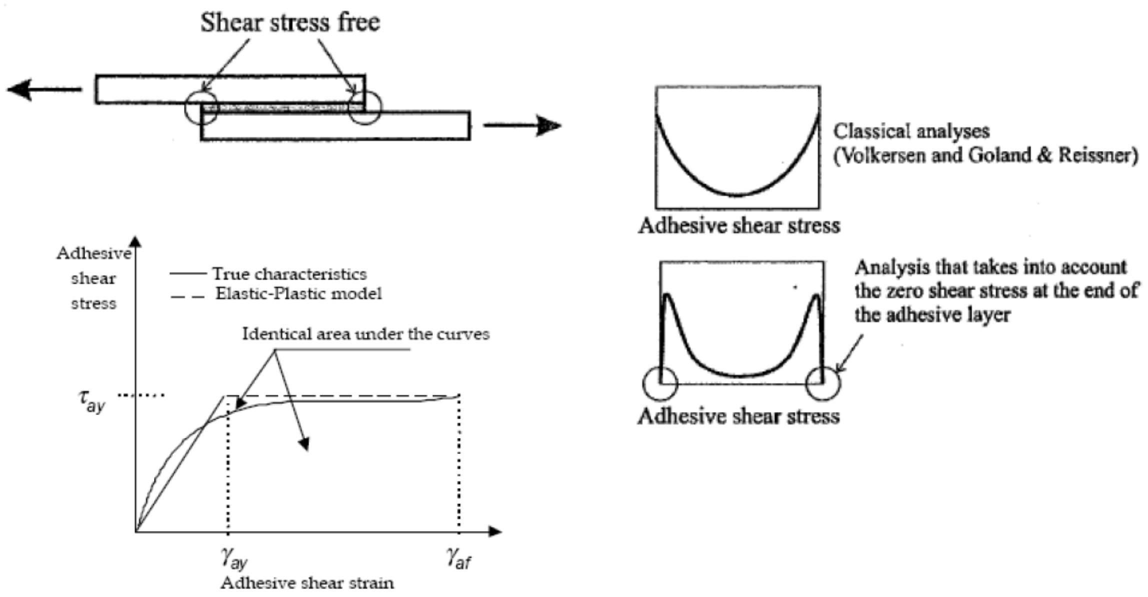
USES 2
Funded by the European Union

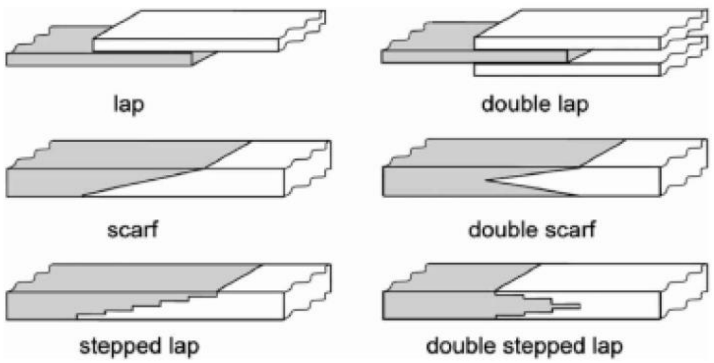
Delamination and bonded joints (ii)

Issues with adhesive joints

- ✓ Difficulties in stress/strength prediction: peeling stresses , non-linear behaviour of adhesive, calibration of the failure criterion (point stress based?, fracture mechanics? as in cohesive elements used in advanced numerical simulation)

- ✓ Unreliable bonding quality (surface preparation, “kissing bonds”), and limited capability to detect defects... leading to <u>tough certification requirements</u>! Demonstration of:
  - Ultimate Load capability
  - Damage no-growth under fatigue loads
  - <u>Residual strength at Limit Loads with the bond surface lost</u>





USES 2
Funded by the European Union
AIRBUS

Bolted joints (i)

By-pass load

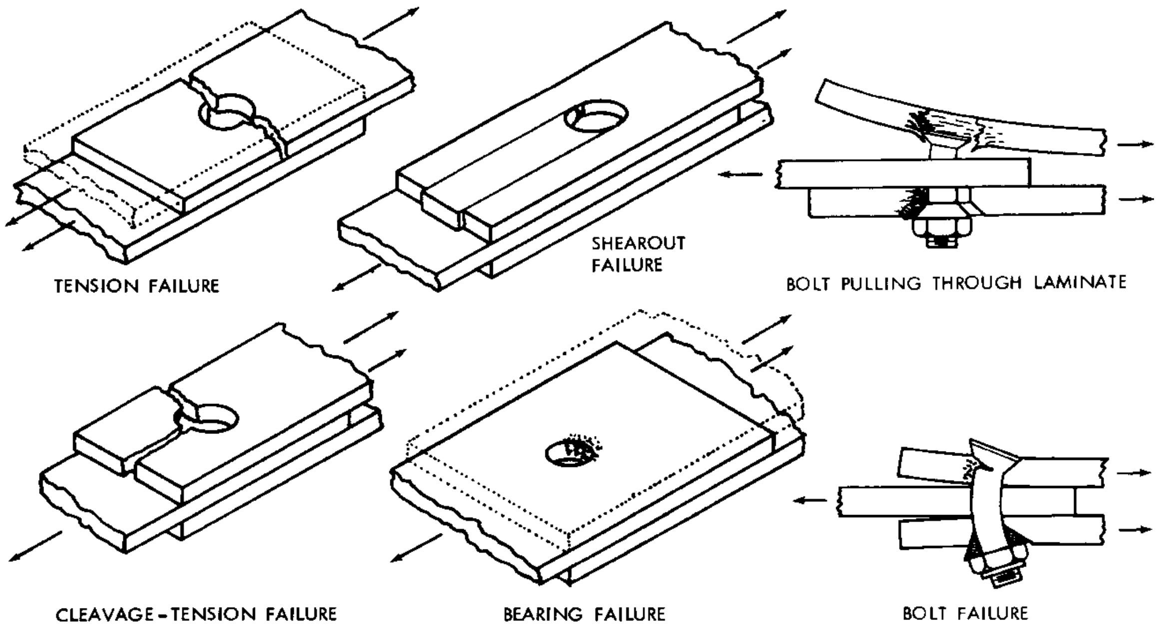
Net section failure under tension or compression by-pass (in-plane) loads

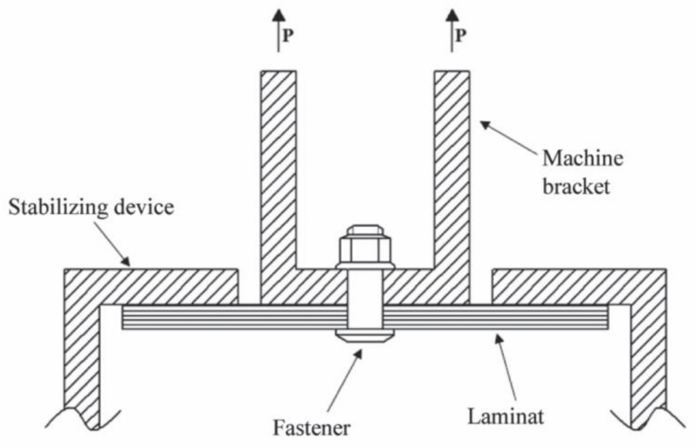
Shear transfer

Bearing, net-section, shear-out, pull-through or bolt failures possible

Pull-out

With bolt in tension, specially with countersunk head rivets, the rivet



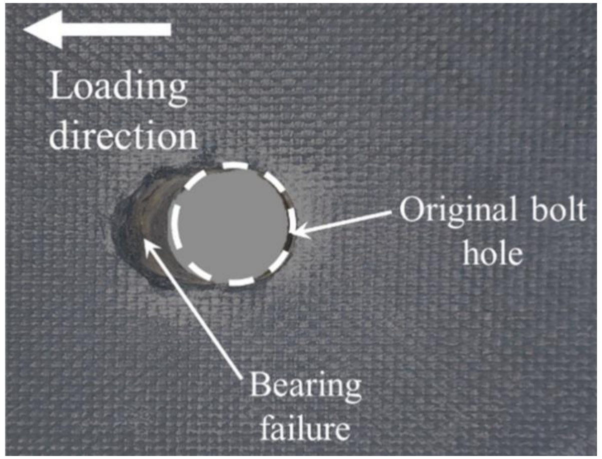


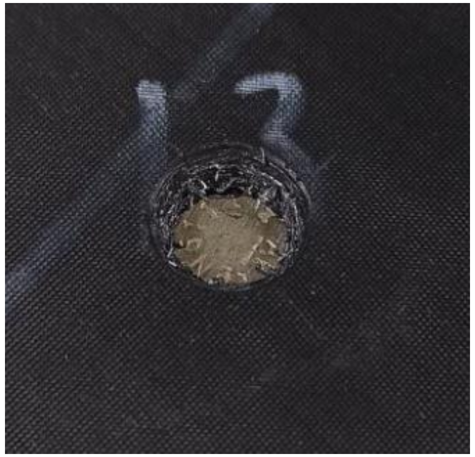
USES 2
Funded by the European Union

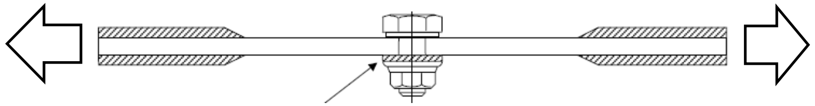
Bolted joints (ii)

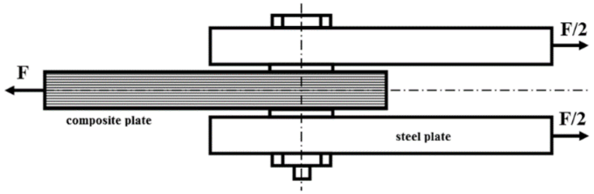
Failed specimens

Filled Hole Tension, Bearing, Pull-Out









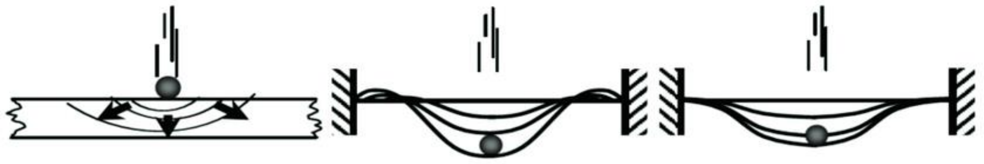
Funded by the European Union

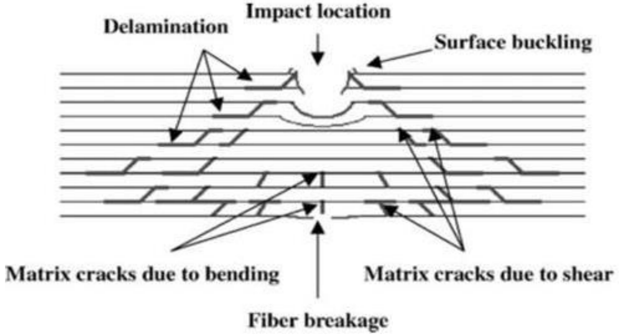
Low velocity impact damages (i)

Barely Visual Impact Damage

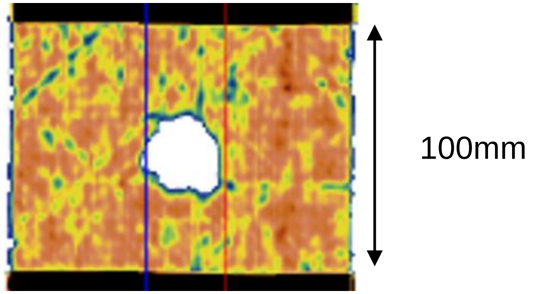
The BVID (Barely Visible Impact Damage) defines the threshold of detectability, during scheduled or directed field inspection, for a damage caused by an low velocity impact, which the part could possibly suffer

- ✓ Possible indications: dents (holes if penetration), fibers broken on surface, cracks or disbond at laminate edges...
- ✓ Risk of passing undetected during visual inspection with internal damage degrading strength, specially in case of impacts by blunt objects





30J impact
dent depth 0.3mm



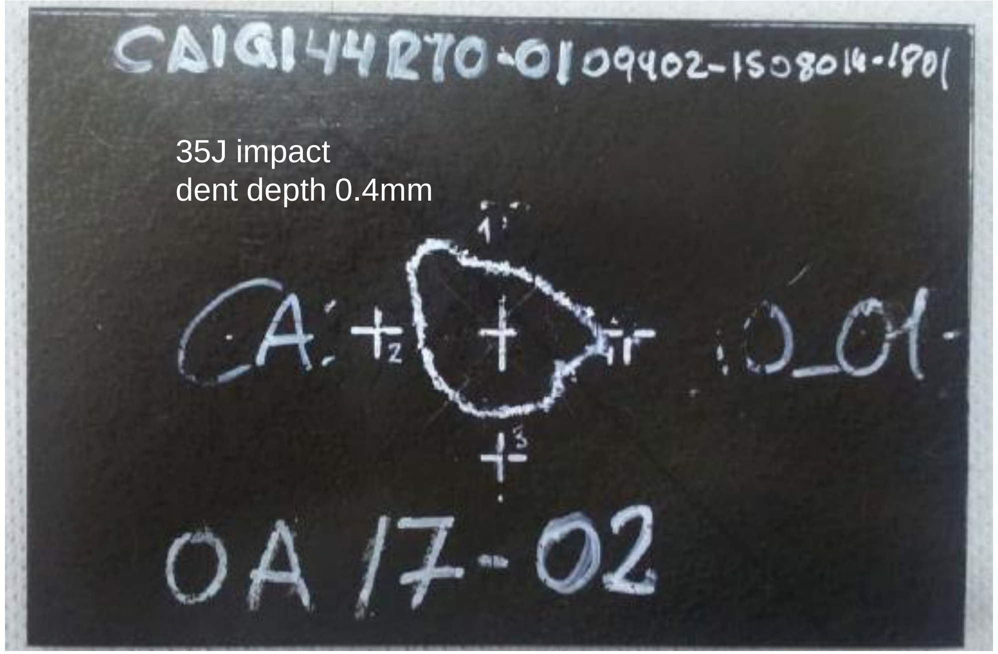
Ultrasonic C-scan

USES 2
Funded by the European Union

Low velocity impact damages (ii)

Residual strength after impact

- ✓ A laminate including a BVID may retain only a 1/3 or 1/4 of its original strength in compression
- ✓ Without sufficiently detectable visual indication
- ✓ But with a internal multilevel delamination of relevant size
- ✓ Some companies estimate these residual strengths from the open-hole configuration
- ✓ Standard practice in Airbus is to obtain design values (ej. CAI, Compression After Impact strength) from dedicated tests campaigns



Funded by the European Union

Fatigue

Highlights

UL capability after Design Service Goal with BVID

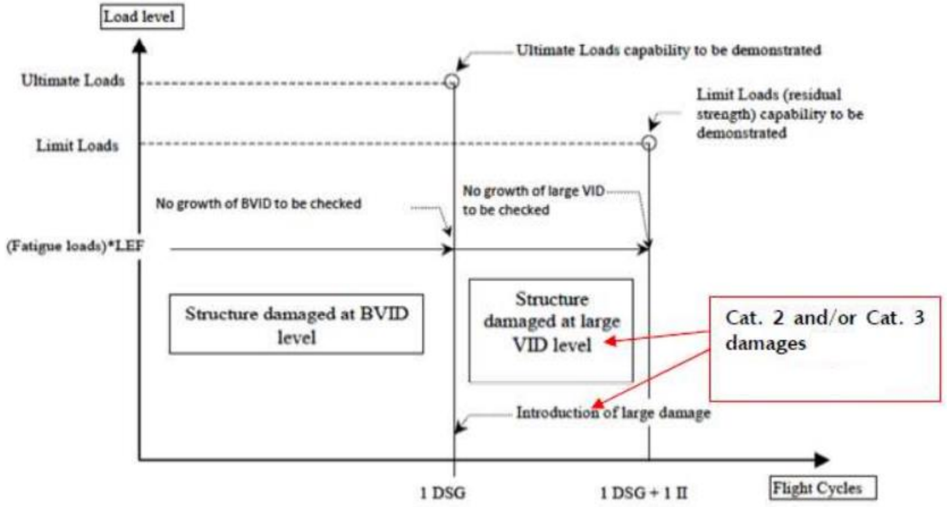
LL capability after Inspection Interval with VID

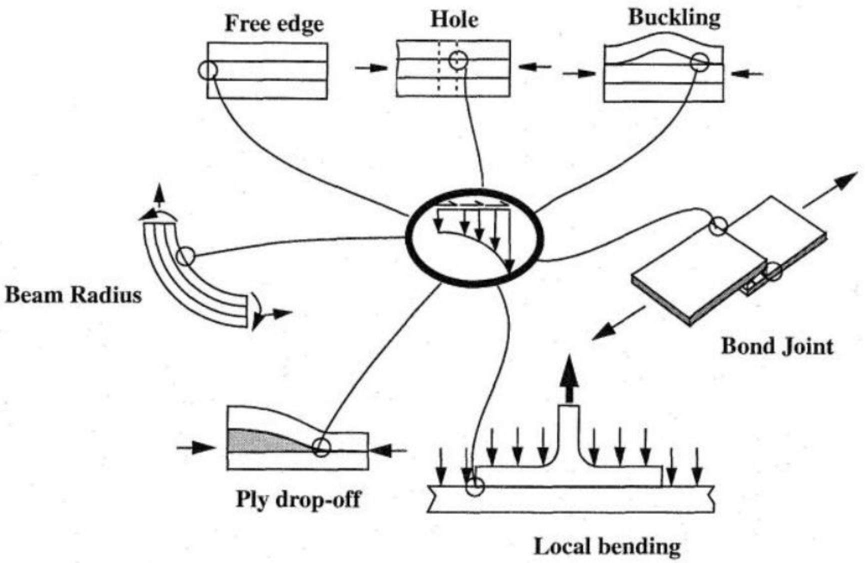
No reliable analytical methods to predict F&DT

No-initiation / No-growth of damages approach (vs stable/slow growth in metals)... Fatigue Insensitive Thresholds

Critical tension/compression cycling (R = -1)

Sensitivity to out-plane loads: risk of onset and growth of delaminations and disbonds





USES 2
Funded by the European Union

Some references

"MECHANICS OF COMPOSITE MATERIALS", 2nd Ed., Robert M. Jones

CMH-17 "Composite Materials Handbook" (Volume 3), 2012 , SAE

"The world-wide failure exercise: Its origin, concept and content" , M. J. Hinton et al, 2004

"Predictive capabilities of nineteen failure theories and design methodologies for polymer composite laminates. Part B: Comparison with experiments ", A. S. Kaddour, 2004

NASA/TM-2003-212663 "Failure Criteria for FRP Laminates in Plane Stress", Carlos G. Dávila, Pedro P. Camanho, 2003

"Composite Laminates Theory and practice of analysis, design and automated layup", S. W. Tsai, J. D. D. Melo, 2017

"Anisotropic Plates", S. G. Lekhnitskii, 1968

"Finite Element Analysis of Composite Materials Using Abaqus" , Ever J. Barbero

USES 2

Funded by the European Union

Thank you for your attention
Questions?

Project 101072599 — HORIZON-MSCA-2021-DN-01

Views and opinions expressed are however those of the author only and do not necessarily reflect those of the European Union or EUROPEAN RESEARCH EXECUTIVE AGENCY. Neither the European Union nor the granting authority can be held responsible for them.

Funded by the European Union